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PERFORMANCE OF THE SPACECRAFT PROPULSION RESEARCH FACILITY DURING ALTITUDE FIRING TESTS OF THE DELTA III UPPER STAGE

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Abstract

The Spacecraft Propulsion Research Facility at the NASA Lewis Research Center's Plum Brook Station was reactivated in order to conduct flight simulation ground tests of the Delta III cryogenic upper stage. The tests were a cooperative effort between The Boeing Company, Pratt and Whitney, and NASA. They included demonstration of tanking and detanking of liquid hydrogen, liquid oxygen and helium pressurant gas as well as 12 engine firings simulating first, second, and third burns at altitude conditions. A key to the success of these tests was the performance of the primary facility systems and their interfaces with the vehicle. These systems included the structural support of the vehicle, propellant supplies, data acquisition, facility control systems, and the altitude exhaust system. While the facility connections to the vehicle umbilical panel simulated the performance of the launch pad systems, additional purge and electrical connections were also required which were unique to ground testing of the vehicle. The altitude exhaust system permitted an approximate simulation of the boost-phase pressure profile by rapidly pumping the test chamber from 13 psia to 0.5 psia as well as maintaining altitude conditions during extended steady-state firings. The performance of the steam driven ejector exhaust system has been correlated with variations in cooling water temperature during these tests. This correlation and comparisons to limited data available from Centaur tests conducted in the facility from 1969 - 1971 provided insight into optimizing the operation of the exhaust system for future tests. Overall, the facility proved to be robust and flexible for vehicle space simulation engine firings and enabled all test objectives to be successfully completed within the planned schedule.

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Introduction

The Boeing Company is nearing development completion of the Delta III, the latest addition to its Delta family of launch vehicles. Among other significant changes from its predecessor the Delta II, the Delta III has a cryogenic upper stage of an entirely new design. Ground tests were conducted to validate the operation of the upper stage systems in a fully integrated fashion, simulating as closely as possible the environmental conditions to which the vehicle will be exposed during flight. The tests included a thermal vacuum soak, demonstration of tanking and detanking of liquid hydrogen, liquid oxygen and helium pressurant gas as well as 12 engine firings simulating first, second, and third burns at altitude conditions.

To accomplish the test objectives set forth, the Spacecraft Propulsion Research Facility (also known as B-2) at the NASA Lewis Research Center's Plum Brook Station in Sandusky, Ohio was fully reactivated for the first time since Centaur upper stage tests were completed in 1971.[1,2] The B-2 facility, shown in Fig. 1, consists of a 65,000 cubic foot thermal vacuum test chamber and an altitude-capable exhaust spray chamber connected by an eleven foot diameter, forty foot long exhaust duct with an isolation valve.[3] Thermal vacuum test conditions were provided by isolating the test chamber and pumping down with mechanical and diffusion pumps to 2 x 10^{-5} torr. Conditions to simulate main engine ignition and operation at altitude were achieved by opening the isolation valve between the test chamber and spray chamber and pumping with a three stage steam driven ejector train.

The test configuration for the present effort differed significantly from the Centaur tests, and therefore provided an opportunity to gain experience in optimizing the operation of the facility over a broader range of upper stage test conditions. Further, several of the facility's capabilities were enhanced during the reactivation period to meet the requirements of the Delta III test plan. Of particular significance were modifications to the propellant and pressurant storage systems and the control and abort systems enhancements.

The purpose of this report is to document the B-2 facility enhancements, integration with the test article, and operational performance during the Delta III upper stage firing tests. Significant differences in the facility configuration and performance from that during Centaur tests are noted as are cases where current test requirements were less than facility capabilities.

Test Article

The test hardware consisted of the upper stage to be tested and the support systems required for its safe and proper operation. These support systems include the facility interfaces which acted to replace the Ground Support Equipment (GSE) that normally supports the preparation of the flight vehicle. In addition to the "GSE" support systems, some additional systems were required to support the operation of the stage in a non-flight configuration (i.e. within a vacuum chamber). These support systems included: the vehicle support structure, engine purges, a low pressure hydrogen vent, a test sequence controller, and an abort system.

Test Vehicle

The vehicle used during these ground tests (the X-Stage) was a slightly modified version of a flight upper stage. Figure 2 shows a photograph of the X-Stage installed in the B-2 test chamber. The major differences between the X-Stage and a flight upper stage were that several non-flight pressurant solenoid valves were used, fixed struts replaced the engine gimbal actuators, the nozzle extension was not used, and extensive non-flight instrumentation was applied.

The upper stage's liquid hydrogen (LH2) tank is located forward of the liquid oxygen (LOX) tank. Each propellant tank is instrumented for pressure, temperature and liquid level. An intertank composite strut structure connects the LOX and LH2 tanks. An equipment shelf below the oxygen tank supports the vehicles avionics.

The stage's main propulsion system consists of a single Pratt & Whitney built RL10B-2 liquid hydrogen/liquid oxygen expander cycle rocket. The RL10B-2 is similar to other models of the RL10 but with an uprated operating condition that requires a modified nozzle profile and includes a large carbon-carbon nozzle extension for increased altitude performance.[4] The nozzle extension was not used during X-stage testing, and in the configuration tested, the engine was rated to nominally provide 23,200 pounds of thrust. The intentions of this test program were to demonstrate normal system operation, therefore for all tests, the inlet conditions of the fuel and oxidizer at the pump inlets were nominal.

Vehicle Propellant Systems

The X-Stage liquid hydrogen system consisted primarily of a tank and a feed duct which provided the hydrogen to the RL10B-2 fuel inlet valve. The X-Stage liquid hydrogen tank was coated with foam insulation. It had a capacity of approximately 10,000 gallons of liquid hydrogen. Tank instrumentation consisted of redundant pressure transducers, a silicon diode temperature sensor rake which extends the length of the tank, a liquid level capacitance probe which was active in the top 10% of the tank, and two point level sensors. The tank was filled from the bottom using a dual function (fill and drain) normally closed ball valve. The tank was vented using a dual function vent and relief valve. The primary vent uses two symmetric non-propulsive vent lines to discharge vent gases overboard. One leg of this vent line is valved off during ground operations. The tank also has a continuous propulsive vent system which is used during coast periods to vent the hydrogen tank and assist in settling the vehicle propellants. The hydrogen tank was initially pressurized using gaseous helium from the on board gaseous helium supply. Once the engine reached steady state operating conditions, the helium pressurant gas was augmented with gaseous hydrogen which was supplied by the engine. The liquid hydrogen feed duct was foam insulated and equipped with a screen to keep any debris from entering the hydrogen turbopump.

The X-Stage liquid oxygen system consisted primarily of a tank and a feed duct which provided the oxygen to the RL10B-2 oxidizer inlet valve. The X-Stage liquid oxygen tank had a capacity of approximately 3400 gallons and was initially uninsulated. After the first multiple firing test, a temporary blanket was fastened to the lower half of the tank for the remainder of the ground tests. The need for this blanket will be discussed later. The LOX tank instrumentation was similar to that in the LH2 tank. The tank was filled using a dip tube and a dual function (fill and drain) normally closed ball valve. The ullage is vented using a dual function vent and relief valve. The vent uses two symmetric non-propulsive vent lines to discharge vent gases overboard. For this test program, one leg of the vent was capped off. The oxygen tank was pressurized using gaseous helium from the on board gaseous helium supply. The liquid oxygen feed duct was an uninsulated line equipped with a screen to keep any debris from entering the oxygen turbopump.

Gaseous helium was primarily used on the X-Stage for tank pressurization, valve actuation, vehicle purges, and engine purges. During test preparations, helium was continuously supplied by the facility to replenish both "GSE" and flight helium consumption. Just prior to the initiation of the autosequence, the facility helium supply to the flight bottles was terminated. Therefore, during testing flight helium was solely supplied by four of the vehicle's five Inconel lined graphite fiber overwrapped storage bottles.

Attitude Control System

Four sets of hydrazine thrusters provide attitude control and propellant settling for the upper stage. For this test program, the testing of the Attitude Control System (ACS) consisted of a single hydrodynamic test using de-ionized water rather than hydrazine.

Vehicle Control Systems

The X-Stage avionics system monitored and controlled all vehicle systems including command of RL10B-2 functions and tank pressurization. The avionics components were mounted on the equipment shelf below the oxygen tank and were protected by a thermal blanket. The primary components of the avionics system were: the Redundant Inertial Flight Control Assembly (RIFCA), the Interface Control Electronic (ICE) assemblies, the Power Switching Assembly (PSA), the Power Contactor Assembly (PCA), the Ordnance Box, the Master Telemetry Unit (MTU), redundant data buses, and redundant battery simulators. The initial software for RIFCA, as well as any subsequent revisions, was operationally verified prior to its installation on the X-Stage at Plum Brook and was verified again at Plum Brook prior to each test.

Facility Systems and Vehicle Interfaces

In order to accomplish the objectives of the X-Stage test program, several critical facility to X-Stage interfaces were required. A means of securing the X-Stage within the test chamber had to be devised. Propellant supplies were required which could simulate the performance of the launch pad umbilicals. Also, autosequenced aborts had to be monitored and implemented independently of vehicle avionics (RIFCA) while keeping in time with RIFCA operations.

Vehicle Support Structure

The X-Stage support structure utilized the major components of the Centaur support structure which still existed at the B-2 facility from the Centaur testing that was performed in the B-2 facility from 1969-1971. The main interface to the X-Stage was through the three columns used for the Centaur stretch unit as shown in the cutaway drawing in Fig. 3. The forward end of the X-Stage interfaced to the facility stretch unit columns through a triangular structure designed and provided by The Boeing Company. Load cells were located at the interface of each column to this triangular structure in order to measure the weight of propellants loaded on the X-Stage. The aft end of the X-Stage was secured to hard points on the test chamber wall in order to restrain lateral motion of the vehicle.

All facility propellant, purge, and instrumentation connections to the X-Stage interface had flexible connections to simplify installation and allow for thermal contraction at cryogenic temperatures. A circular work platform was constructed around the vehicle that gave access to the upper levels of the hydrogen tank.

Facility Propellant Systems

Liquid hydrogen was supplied to the X-Stage from a 34,000 gallon hydrogen storage dewar located approximately 250 ft from the facility. A 3 inch vacuum jacketed line provided liquid to just outside the test chamber. A 4 inch globe valve controlled liquid hydrogen flow through 4 inch foam insulated lines that were connected to the hydrogen tank fill through the GSE interface on the vehicle umbilical panel. The facility hydrogen supply was filtered both outside and inside of the test chamber. The 6 inch facility hydrogen vent line interfaced to the hydrogen tank non-propulsive vent lines to capture all non-propulsive vent gas. The propulsive vents discharged small amounts of hydrogen into the test chamber. These vents were only active during vehicle coast mission phase when the test chamber was below 1 psia and the steam ejector system was pumping on the test chamber. A 19,500 gallon liquid hydrogen dewar, known as the hydrogen dump tank and located in the spray chamber below water level, was used for off-loading hydrogen from the vehicle.

Liquid oxygen was stored in a 12,000 gallon dewar located 150 feet from the facility. A two inch foam insulated line ran to a 4 inch butterfly valve which was used to control flow to the X-Stage. A 4 inch foam insulated line penetrated the test chamber and was connected to the oxygen tank fill through the GSE interface on the umbilical panel. The facility oxygen supply was filtered both outside and inside of the test chamber. The 6 inch facility oxygen vent line interfaced to the single oxygen tank vent line in order to capture all vent gas. During chilldown of the turbopumps, oxygen flowed through the LOX turbopump and was discharged through the combustion chamber of the engine into the B-2 test chamber. The oxygen was removed by the main or auxiliary ejectors which ran continuously during the oxidizer pump chilldown. A 6,000 gallon liquid oxygen dewar located in the spray chamber below water level served as the oxygen dump tank for off-loading vehicle oxygen.

A new facility 5,500 psig helium supply was regulated and used to charge the flight helium bottles through the GSE interface on the umbilical panel prior to launch simulation. The facility provided the "GSE" helium supply which was used primarily for the actuation of the LOX and LH2 fill and drain valves and vent and relief valves. The "GSE" helium interfaced with the X-Stage at the normal GSE interface on the umbilical panel. This helium supply was not terminated during vehicle operation as it normally would be for flight. It was available as needed for system operations in the event of an abort.

In addition to the "GSE" helium, the facility also supplied both helium and nitrogen purges to the engine. These purges are not used in flight but must be provided to protect the critical engine components from the water vapor present between tests. These purges are standard for ground operation of an RL10 engine. They connected directly to the engine as required and did not impact normal engine operation. Particular attention was paid to the flow rates and timing of engine purges to minimize their affects on temperature conditioning of the engine's turbomachinery.

Test Chamber Vacuum System

The test chamber was designed to be isolated from the spray chamber and pumped to achieve a vacuum of 5 x 10^{-8} torr under clean, dry, empty conditions. The vacuum system consists of a three stage mechanical pumping system and ten oil diffusion pumps. Four 728 CFM roughing pumps make up the third stage of the mechanical pumping system. Two 1875 CFM blowers and one 28,100 CFM blower, make up the second and first stages of the mechanical pumping system respectively. High vacuum is achieved using the ten 35 inch oil diffusion pumps.

Facility Altitude Exhaust System

The altitude exhaust capability of the B-2 facility is provided by two parallel three stage steam driven ejector trains. For these tests, a single three stage branch of the main ejector had sufficient pumping capacity to maintain altitude conditions and reduced the main ejector steam consumption. Two other steam driven ejector systems that support the altitude capability are the auxiliary ejectors and the Low Pressure Vent. The auxiliary ejectors have less capacity than the main ejectors but can slowly evacuate the spray chamber while consuming less steam than the boilers supply for charging the accumulators. Thus, the spray chamber can be evacuated or maintained at vacuum without depleting the stored steam supplies.

The Low Pressure Vent (LPV) is an existing facility steam driven ejector that was designed to safely remove the hydrogen used for RL10 turbopump thermal conditioning from the facility. The hydrogen vented through both cooldown valves as well as the hydrogen gas from other engine vents all tied into the facility low pressure vent. The engine was protected from the possibility of steam backstreaming by the use of an autosequenced abort and an isolation valve which was controlled by the abort system.

Engine Exhaust Duct

The B-2 facility was designed to remove the exhaust of the Centaur's two early model RL10 engines from the test chamber through the 11 foot diameter exhaust duct. The structural integrity of the exhaust duct is maintained against the hot exhaust gases by a back side water cooling spray. A concern for this test program was that the original facility design did not provide direct backside spray cooling of the upper 12 inches of the duct. This was not an issue for the Centaur tests, because the vehicle was installed such that the exits of the engine bells were 12-18 inches below the exhaust duct inlet and exhaust impingement was on a well cooled portion of the duct wall. However, because the installation of the X-Stage used existing structure, the structure stack up left the engine exit 48 inches above the inlet to the exhaust duct. Analysis and empirical information from previous firings of the RL10B-2 led to a prediction of exhaust impingement 18 inches below the duct inlet. However, the prediction had a large uncertainty associated with it. To resolve this issue, an uncooled duct liner/extension was fabricated from a high temperature steel. This duct insert, which was considered sacrificial for these tests, extended from 18 inches below the exhaust duct inlet to 18 inches above the duct and was 6 inches smaller in diameter than the exhaust duct.

Facility Control and Abort System

A new facility supplied control system that was used for these tests was a redundant programmable logic controller (PLC) system. The two PLC's operated in parallel and acted not only as the sequencer and abort system, but also as its own watchdog computer. The sequencer portion of the system controlled the required facility valves (primarily engine purges) throughout the test sequence. The abort portion of the system monitored 17 automatic abort parameters shutting down and securing all of the systems which it controlled as required in the event of an abort. The redundancy of the PLC's allowed for a convenient means to monitor the PLC health. Each sequencer periodically outputted a signal to act as a "watchdog" signal. The abort system watched for this signal, if it would have been interrupted in any way the sequence would have been terminated. The abort system was designed to fail safe upon loss of power or loss of any signal which was being monitored as an abort.

The sequencer was operationally verified at two levels prior to the start of testing. The lowest level check-out did not actuate any valves, but simulated a run within the control room only. At the next level, a dry run check out was conducted which actuated valves, but without propellants in the test package. The dry run was typically performed as an integrated check-out with the vehicle's systems.

RIFCA commanded the X-Stage systems during a test. The PLC's operated in parallel with RIFCA to coordinate any required "GSE" and facility valve operations as well as to monitor all abort parameters. The PLC's stayed synchronized with RIFCA by receiving periodic event markers such as the engine start signal. During a normal shut down, RIFCA and the PLC's both shut down independently at the end of the sequence. In the event of an auto-sequenced or manual abort, the abort bus removed power from the RL10B-2 engine solenoid bus as well as the ordnance bus to terminate the test in a safe manner. All sequenced valves went to fail safe positions. In addition, the X-Stage LOX and LH2 tank vents were driven open.

Test Description

The tests that were conducted are listed in Table I with a brief description of each test. The test number represents tests one through eight from the original test plan. Although there were some differences from the original test plan, these numbers are kept here for consistency with other documentation which may reference the test plan. For example, test one was conducted as two separate tests (1a and 1b) because it simplified test scheduling, and test 4, a two burn mission simulation, was eliminated by combining its objectives into tests 3 and 5. Aborted tests and the cause for the abort are noted in the comments section of the table.

In addition to facility checkout tests, eleven tests were conducted with the X-Stage installed. Three of these tests were aborted but were successfully repeated to achieve all test objectives. The engine was hot-fired twelve times with single burn durations ranging from eleven to 177 seconds and a maximum multi-burn accumulative time of 248 seconds. Each burn was followed by a simulated coast period and a subsequent turbopump chilldown procedure. In order to conserve the facility's steam supply, the burn durations were typically reduced from the actual mission durations and the main ejector train was shutdown during simulated coast periods.

Objectives

As was stated in the introduction, the purpose of these ground tests was to validate the nominal operation of the Delta III upper stage systems in a fully integrated fashion, simulating as closely as possible the environmental conditions to which the vehicle will be exposed during flight. The following global objectives were used to establish a test plan which would achieve this validation:

1. Verify the physical and functional integrity of Delta III second stage systems under hot firing conditions.

2. Demonstrate Attitude Control System thermal performance and hydrodynamic characteristics.

3. Correlate second stage thermal models for propellant loading, engine firing and coasts.

4. Verify the absence of POGO.

5. Determine engine induced vibration environments.

Several unnumbered facility checkout tests were conducted prior to the arrival of the X-Stage for two purposes. First, the past performance of the altitude exhaust system was not well documented, and these tests provided data on steam consumption, pumping rates and capacity, and the cooling water system configuration. Second, the tests provided experience for the system operators.

Procedures

Several days before a test, the facility gas and cryogenic systems were charged and filled to acceptable levels. On the day prior to a test, the steam accumulators were charged to their full capacity and a final inspection of the test article was performed to verify its readiness for testing. Once the test article and test chamber passed inspection, the test chamber was sealed and pumped down. Early on the test day, pump down of the spray chamber was initiated with the auxiliary ejectors. The auxiliary ejectors were used because they did not deplete steam from the accumulators.

The next step was to power up the vehicle and verify its function electrically. This was followed by initiation of pressurizing the vehicle helium bottles to 4500 psig. In parallel with the helium pressurization, propellant loading was initiated. First the facility LOX lines and dump tank were chilled down to liquid temperatures. The vehicle LOX fill and drain valve was then opened, and the facility control valve was used to flow slowly to the vehicle tank. Once the tank was approximately ten percent full, the oxygen flow rate was increased to fill the tank more quickly. When the tank approached its desired fill level, the flow rate was reduced so that a more accurate final level could be achieved. After oxygen flow to the vehicle was established, the hydrogen tank fill was initiated through a similar procedure.

With propellant tanks loaded to their desired levels, the Active Pressure Control systems were verified. After the pressurization system tests, the tank levels were checked and topped off as necessary. When top-offs were complete the facility propellant lines were drained and purged.

At this point final facility preparations were made and the eleven foot valve isolating the test and spray chambers was opened. The main ejectors were only operated from one minute before to one minute after a firing to conserve steam. Once the facility preparations were completed, the RIFCA flight program was initiated, simultaneously sending a synchronization signal to the facility PLC's. From then on, manual operations were limited to operating the main ejector train and monitoring the 18 manual abort parameters. RIFCA controlled the turbopump chilldown and engine start, burn duration, and coast activities.

When the test was completed, the test conductor proceeded to facility clean up. Propellants were transferred from the vehicle tanks to the dump tanks, facility systems were inerted and secured as required, and the test chamber and spray chamber were repressurized. It was then possible to inspect the test article on the morning following a test.

The only major deviation of this process occurred for the ascent profile tests. If the test to be performed involved an ascent profile simulation, the test and spray chambers had to be inerted. This process began the day before the test, when the test chamber and spray chamber were evacuated. The spray chamber was repressurized to ambient pressure with gaseous nitrogen and left overnight. On the day of the test, the spray chamber received an additional vacuum purge and was repressurized to approximately 13 psia using gaseous nitrogen while the test chamber remained isolated. The test chamber was left under vacuum until the vehicle was fully tanked, at which point the test chamber pressure was equalized with that of the spray chamber. The 11 foot valve was opened and the main ejector train was brought on line. Both chambers were then evacuated to below 1 psia prior to the start of the hot fire.

Facility Performance

During testing, the performance of each of the B-2 facility's systems was monitored to ensure that the proper environment was being provided for the vehicle to complete its test objectives. The facility's systems successfully satisfied the test requirements.

Facility Propellant Systems Performance

The propellant flow rates provided by the facility fill lines are summarized in Table II. The values provided were typical of all fill and drain procedures conducted during the test program. These flow rates were calculated from the vehicle weight measurements obtained from the support structure load cells. The hydrogen flow rates determined in this manner were verified against flow measurements taken with a facility orifice. The oxygen system did not have a flow meter. The basic test requirements were to supply low and high fill rates approximately simulating the expected fill rates at the launch pad. The flow rates were controlled by the storage dewar pressure and by adjusting existing control valves located just outside the test chamber. The low flow rate was used initially to chill the tank down and for final top-off of the level. The high flow rate was used for the majority of the fill. In both the hydrogen and oxygen cases the high flow rates used were not facility maximums, but were sufficient for the vehicle fill operations.

Tank draining was accomplished by flowing back through the fill line out of the test chamber, and down into the dump tanks. The maximum tank draining flow rates are also shown in Table II and were limited by the vehicle fill and drain line size and the maximum operating pressure of the vehicle tanks.

Test Chamber Vacuum System Performance

The first vehicle test conducted was the thermal vacuum test, which was intended to provide anchoring data for thermal model development. To minimize the convective heat transfer to the vehicle, it was desirable to evacuate the test chamber to less than 10^{-4} torr. For this test the vacuum chamber was isolated from the spray chamber and mechanically evacuated. Figure 4 shows the pressure pumpdown of the test chamber with the mechanical vacuum system. The numbers on the plot indicate when various pumps were activated. Pumps 1 through 10 are oil diffusion pumps, pumps 11 through 13 are blowers and the remaining pumps are mechanical pumps. The test cell was deadheaded at a pressure of $2x10^{-5}$ torr satisfying the test requirements. The gap in the data near 10^{-3} torr was between the effective ranges of two instruments.

Facility Altitude Exhaust System

The performance of the main ejectors was not well documented prior to these tests. Because of this, predictions of steam consumption were based on the original manufacturer's specifications. These predictions are compared to data collected during testing in Table III. In general the predicted steam consumptions were comparable to the test data. The differences may be attributed to uncertainty in the numbers and operating the facility steam ejectors at different conditions than originally specified. Conditions that were different included slightly higher steam supply pressures, non-chilled intercondenser spray water, and lower intercondenser water flow rates. However, for the current engine, steam supply and ejector configuration, engine operating durations up to 650 seconds could be accommodated.

The altitude exhaust system utilizes water spray in the spray chamber and between ejector stages to condense both water vapor from the engine exhaust and the steam used to drive the ejectors. The performance of the condensing sprays is a function of the cooling water temperature. The spray chamber holds 1.75 million gallons of cooling water. To enhance its operation, the facility was built with a water refrigeration system designed to cool the spray chamber water to 40 °F before a test. The refrigeration system was not reactivated for the X-Stage tests. Therefore, the spray water was cooled by pumping it into an outdoor retention pond between tests to cool (the water temperature typically warmed less than 10 °F during a test). A disadvantage of this technique was that the weather controlled the temperature of the spray water, and an early spring in northern Ohio meant that the last firing was conducted with 68°F water. One advantage was that data was collected on the exhaust system performance over a range of water temperatures, without compromising the test objectives. Figure 5 summarizes the spray chamber pressures achieved both with and without the engine firing as a function of water temperature. Reducing the water temperature significantly reduces the pressure in the spray chamber. Also included on the plot are operating points from the exhaust system design curve and an estimate of the performance during Centaur tests. There is agreement between the current and Centaur test data, and both tests showed performance better than the design curve.

In addition to removing the non-condensable RL10 exhaust from the spray chamber, the main ejectors were used to approximately simulate the boost phase ascent pressure profile that the upper stage will experience in flight. Figure 6 compares the B-2 facility pumpdown during test 3 with an estimated flight pressure profile. It took about three times as long to reach to 0.5 psia during the test as it would in flight. The slower pumpdown rate was expected because the pumpdown time is a function of the large volumes to be evacuated and the capacity of the ejector system. The time for the simulation could have been significantly reduced (by a factor of two) by operating both branches of the main ejector in parallel, however, for the X-Stage test objectives the slower pumpdown rate was acceptable.

Engine Exhaust Duct

Thermal Performance. During engine firings, the 11 foot exhaust duct and the duct extension were exposed to the severe thermal environment of exhaust impingement. Several thermocouples were attached to the backside of the uncooled extension and to the inside surface of the cooled duct to monitor the temperature (and structural integrity) of these components. The highest temperatures measured on the extension and the duct during the first firing (test 3) are plotted in Fig. 7. Both temperatures rose rapidly, raising concern for the structural integrity of these components as test durations increased.

A thermal analysis was conducted with data from the first three firings (one firing from test 3 and two firings from test 5b) to extrapolate the data of the cooled duct to the longer duration firings planned for tests 6 and 7. Considering all the available exhaust duct thermocouple data, two issues became apparent. First, there was significant circumferential variation of the heat transfer to the duct, and second, there was significant run to run variation in the heat transfer to the duct. The highest temperatures measured on the cooled exhaust duct for each firing were used in the analysis.

The first step of the analysis was to calculate heat transfer coefficients for both the hot exhaust gas side and the water spray side of the cooled duct. The two heat transfer coefficients were then used to calculate an equilibrium duct wall temperature. According to the analysis, at the equilibrium temperature, the heat flux from the hot gases should have been balanced by the cooling from the water spray. The predicted equilibrium wall temperatures ranged from 520 to 850 °F for the three firings analyzed. These temperatures were well within the limits of structural integrity for the duct material. Figure 8 presents the highest temperatures measured during testing (test 6b). The third burn of test 6b was a longer duration burn, and while equilibrium had not been reached by engine shutdown, the temperature of the cooled duct was still within the predicted equilibrium temperature range. However, an uncalibrated infrared camera view of the cooled exhaust duct indicated some local hot spots approximately three feet below the duct inlet. Since these hot spots were not near any of the instrumentation, their temperatures could not be determined. Figure 8 also includes data from the uncooled duct extension during this burn and shows that the temperature of the extension exceeded 1500 °F. Since this extension was considered sacrificial, the high temperatures were not a test concern. A post-test inspection of the duct extension was conducted and identified only minimal (less than 1/8 inch) local warping of the extension and some weld cracks. Some thermal discoloration was sustained by the cooled duct in the location of the local hot spots.

Diffuser Performance. As was expected, the 11 foot exhaust ducted acted like a diffuser and pumped the test chamber to a lower pressure than the spray chamber. This behavior created a differential pressure between the two chambers of up to 0.7 psia. The pumping behavior during test 5b is shown in Fig. 9. As the plot indicates, the pressure in both chambers was reduced to about 0.25 psia prior to ignition by the main ejector train. During the firing, the test chamber pressure was reduced to 0.03 psia while the spray chamber pressure was increased to 0.5 psia. The test chamber and spray chamber pressures ranged between 0.03 and 0.15 psia and 0.49 and 0.79 psia respectively during the 12 firings. An autosequenced abort upper limit was set on spray chamber pressure at 1.0 psia, which provided margin below the pressure at which the exhaust duct would unstart.

At engine shutdown, a rapid back flow from the spray chamber to the test chamber equalized the pressure difference. The flow into the test chamber was mostly hot steam and carried water spray droplets. The backflow had several undesirable effects. First, the steam formed frost on many of the vehicle and engine's cold components. Second, the impingement of the steam on the LOX tank added heat to the liquid oxygen. Temporary insulation was added to the lower half of the oxygen tank to resolve this problem. Third, the electrical connections had to be protected from the water. The backflow of steam was expected, as it had occurred during Centaur tests. However, differences in the installations of the two vehicles and facility operation made the pressure equalization during X-Stage tests much more rapid than during Centaur tests.

Two key differences between the test setups contributed to the stronger shutdown backflow. The X-Stage installation had a larger flow area between the spray and test chambers, and during the Centaur tests, the cooling water refrigeration system was used which reduced the spray chamber pressure. Figure 10 shows the Centaur installation with a work platform and flexible boot around the engines. The platform restricted the flow between the two chambers and deflected it away from the vehicle. During X-Stage tests, there were no restrictions in the 11 foot duct. Although no high speed pressure data is available from the Centaur tests, a typical comparison of the test chamber pump down after engine start for the two tests is shown in Fig. 11 and indicates the effectiveness of the flow restriction in slowing the test chamber pump down after engine ignition. Even though Centaur, with two engines, had greater pumping power, the test chamber pressure took three times as long to drop below 5 torr due to the flow restriction. It can be expected that the flow restriction would have a similar effect on the shutdown backflow.

The shutdown backflow did not inhibit the completion of any test objective during X-Stage testing. However, comparison of the data from X-Stage and Centaur tests has shown that, if for a future test the backflow became a concern, several simple modifications to the configuration and operating procedures can greatly reduce the backflow. By refrigerating the spray chamber water to 40 °F prior to a test, the differential pressure between the two chambers during X-Stage testing could have been reduced by 0.25 psi. A further reduction of 0.15 psi could have been achieved by operating both ejectors in parallel. These procedural changes combined with a flow restriction similar to that used for Centaur tests, would minimize the effects of the shutdown backflow.

Facility Control and Abort System Performance

In general the facility PLC's interfaced with the X-Stage electronics as planned. The PLC's monitored a large number of critical abort parameters to ensure the safe operation of these tests, without encumbering their completion. Three tests were aborted prior to completing the test objectives and are indicated in the test summary table.

Test 5a was terminated manually prior to initiating the autosequenced portion of the procedures due to a helium regulator failure. The cause of the regulator failure was determined and the regulator replaced prior to test 5b. The autosequencer initiated an abort of test 6a during the second burn start transient due to a slower than expected acceleration of the engine. The slow acceleration was detected by the Lo-Lo Pc abort as a late rise in combustion chamber pressure. Review of the data indicated that the engine was healthy, and that the RL10B-2 under similar conditions historically behaved similarly. The timing for the Lo-Lo Pc abort was extended to allow for a later acceleration and the test was repeated as test 6b. The final abort was initiated by the autosequencer early in the first burn of test 7a on a high combustion chamber pressure. A quick review of the data determined that this was a false abort due to a bad data channel. Because little steam had been consumed during the aborted run, it was possible to make a second, and successful, attempt on the same day.

Summary

Flight simulation ground tests of the Delta III upper stage were successfully conducted in the B-2 facility at NASA Lewis's Plum Brook Station. These were the first complete stage tests conducted in the facility since it was put into a standby status 25 years ago, and many of the systems had to be reactivated. The tests included thermal vacuum simulation, propellant tanking and detanking, and several flight simulations with a total of twelve engine firings. Despite the lack of recent operational experience, the performance of the facility systems enabled successful completion of all planned tests within the allotted test schedule.

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1. Lewis Research Center Staff, "Centaur Space Vehicle Pressurized Propellant Feed System Tests," NASA TN D-6876, October, 1972.

2. Groesbeck, W.A., Baud, K.W., Lacovic, R.F., Tabata, W.K., and Szabo, S.V., "Propulsion Systems Tests on a Full Scale Centaur Vehicle to Investigate 3-Burn Mission Capability of the D-1T Configuration," NASA TM X-71511, February 5, 1974.

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4. Santiago, J.R., "Evolution of the RL10 Liquid Rocket Engine for a New Upperstage Application," AIAA-96-3013, presented at the AIAA/ASME/SAE/ASEE 32nd Joint Propulsion Conference and Exhibit, July 1-3, 1996.

TABLE I. TESTS CONDUCTED IN THE B-2 FACILITY FOR THE DELTA III UPPER STAGE

Test	Description	LOX Fill Level	LH2 Fill Level	Date	Comments
	Facility Checkouts	NA	NA	11/12 - 12/23/97	Main steam ejectors, water pumps, and steam system tested to determine proper configuration and steam consumption
1a	Thermal Vacuum Avionics Test	NA	NA	2/19/98	Vehicle powered up under high vacuum to provide data for thermal modeling effort
1b	LH2 Fill & Drain Test	NA	97%	2/23/98	Ascent pressure profile simulation, RL10B-2 turbopump spin-up, tank heat leak, tank top off
2	LOX Tank Fill & Drain Test	97%	NA	2/25/98	Tank heat leak
3	Ascent Profile and 60 second burn	95%	95%	3/6/98	
5a	Two burn simulation	36%	39%	3/12/98	Test manually aborted during tank pressurization checkout due to failed pressure regulator
5b	Two burn simulation	36%	39%	3/18/98	
6a	Three burn mission with LOX Depletion	36%	90% min	3/24/98	Abort during second burn start due to low transient chamber pressure (slow acceleration)
6b	Three burn mission with LOX Depletion	36%	90% min	3/27/98	LOX depletion successful, shut down on low chamber pressure
7a	Three burn mission with LH2 Depletion	90%	39% min	3/31/98	Abort on first burn due to false combustion chamber pressure signal
7b	Three burn mission with LH2 Depletion	90%	39% min	3/31/98	LH2 depletion successful, shut down on low fuel venturi pressure
8	ACS Propellant Feed System Water Hammer	NA	NA	4/2/98	

TABLE II. TYPICAL PROPELLANT FLOW RATES DURING X-STAGE FILL AND DRAIN PROCEDURES

Procedure	Low Rate (gpm)	Max Rate* (gpm)	Comments
LOX Tanking	25	158	at max. rate, valve 50% open and storage dewar at 37 psia
LOX Detanking	NA	102	
LH2 Tanking	100	320	at max.rate, valve 50% open and storage dewar at 29 psia
LH2 Detanking	NA	272	

*Maximum rates represent maximum flow rate during test, not a facility limit.

TABLE III. STEAM CONSUMPTION OF FACILITY EJECTORS

Ejector	Predicted Consumption (lb/sec)	Demonstrated Consumption (lb/sec)
Low Pressure Vent	33	25
Main Ejector First Stage	58	50
Main Ejector Second Stage	57	40
Main Ejector Third Stage	60	68
Total	208	183 ± 12

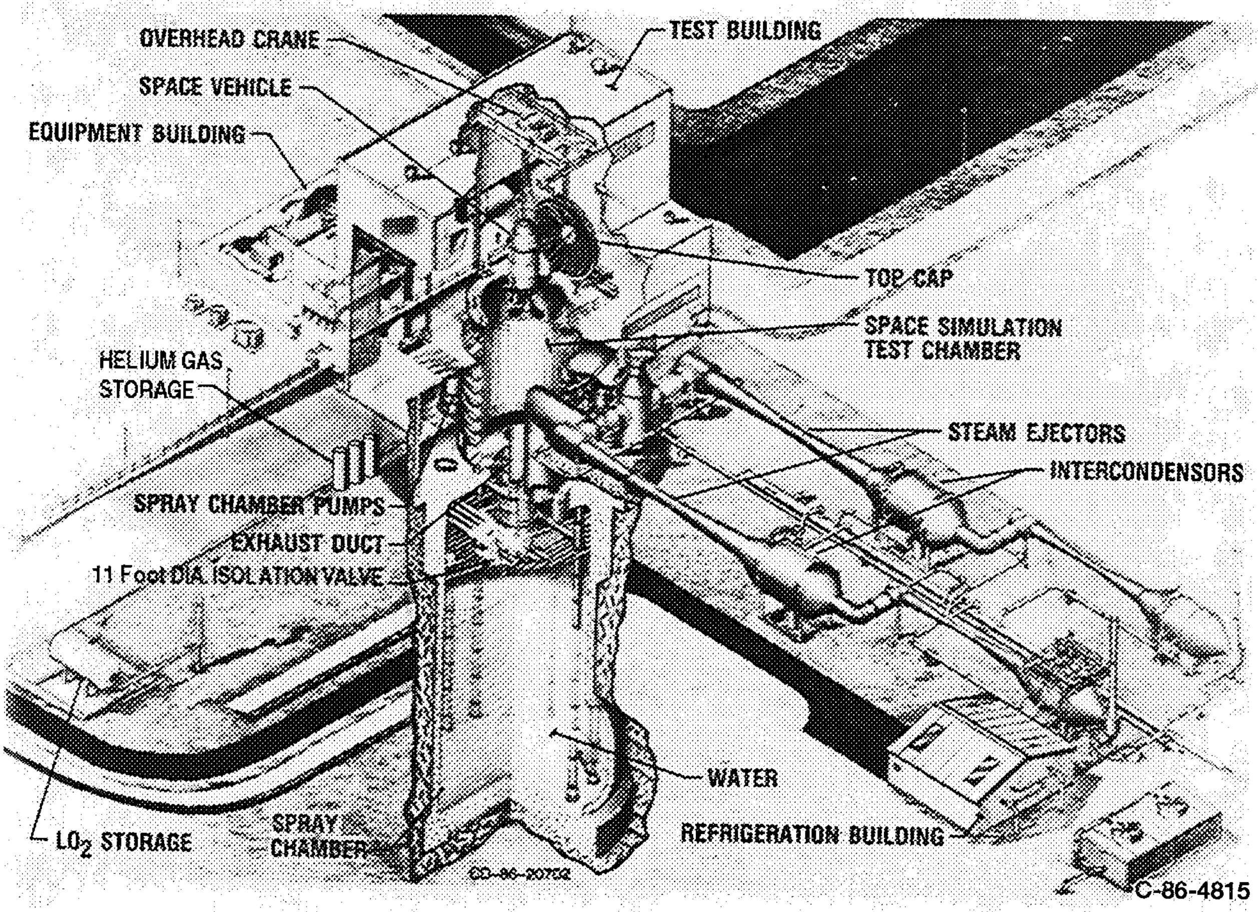

Figure 1.—Cutaway view of the Spacecraft Propulsion Research Facility (B-2).

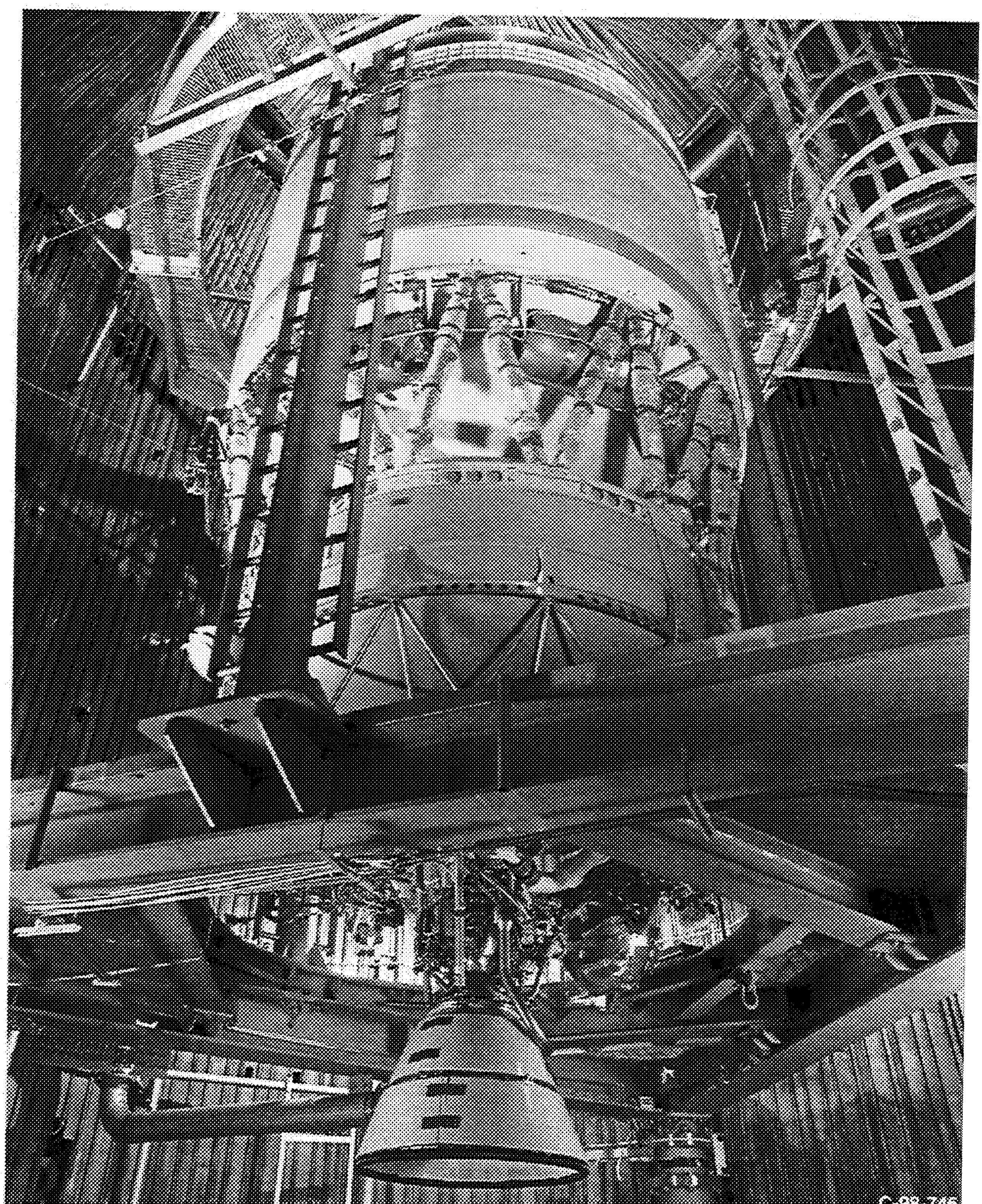

Figure 2.—Photograph of the X-Stage test vehicle installed in the B-2 facility.

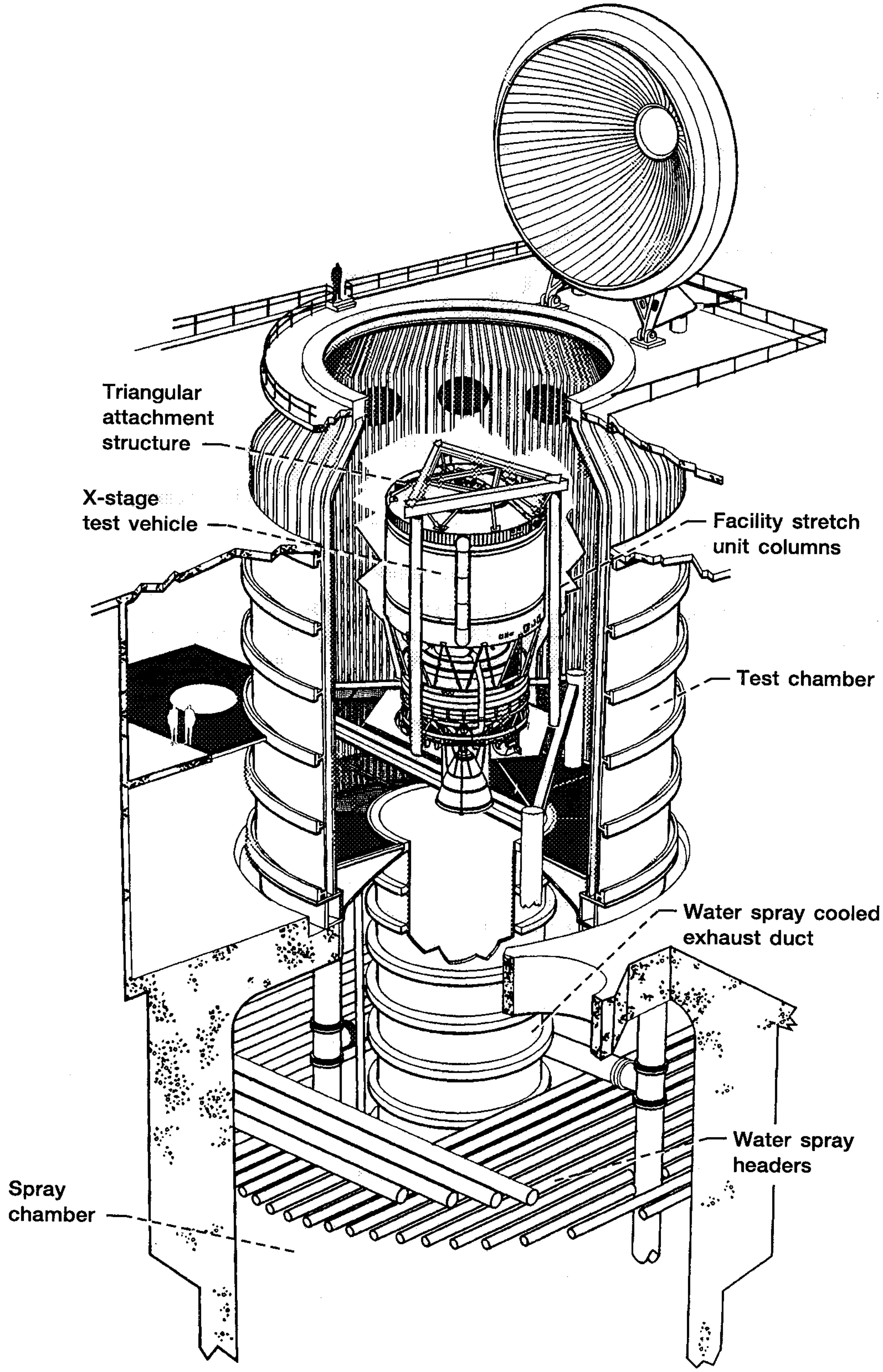

Figure 3.—Cutaway drawing of the X-Stage in the B-2 test chamber.

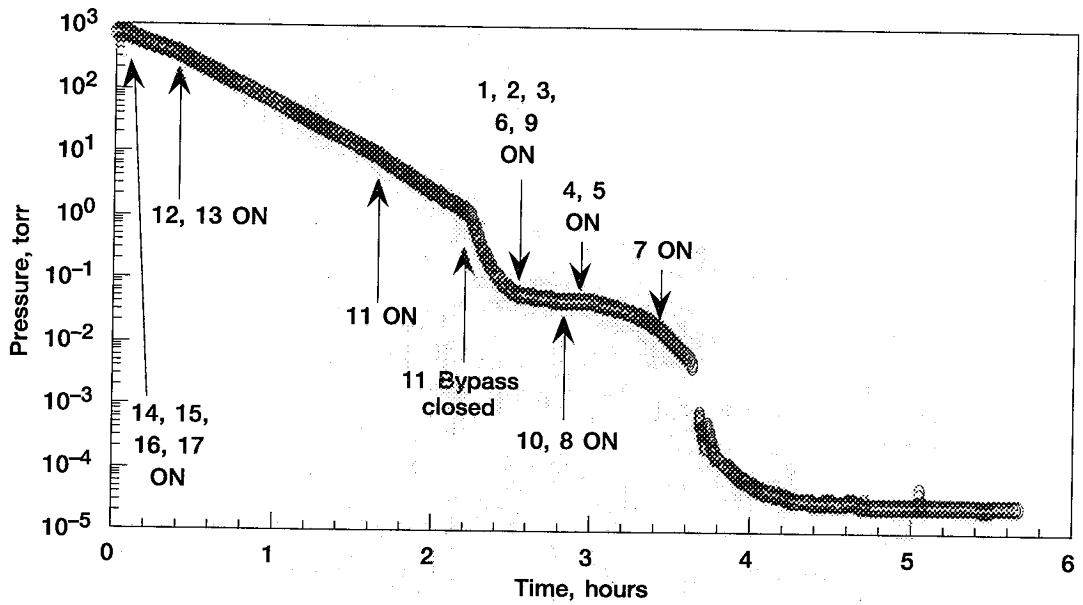

Figure 4.—Test chamber pressure pumpdown for the thermal vacuum test, Test 1a, using mechanical pumps (14-17), blowers (11-13), and oil diffusion pumps (1-10). Note: Data is assembled from 3 pressure transducers with different ranges.

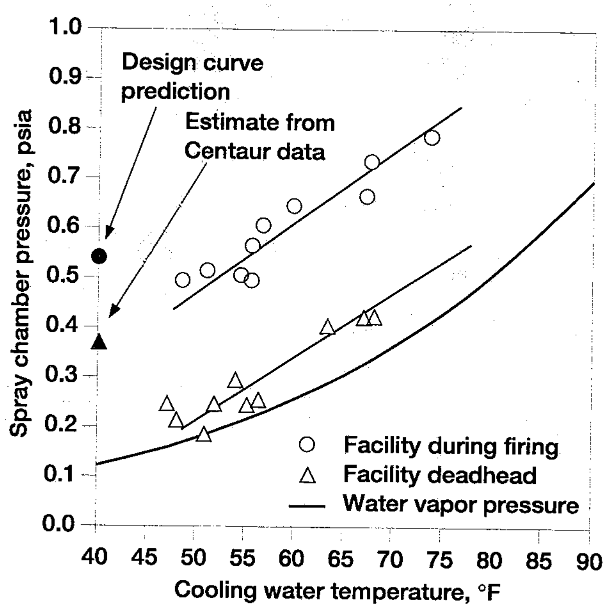

Figure 5.—Spray chamber pressure before and during X-Stage firings with the main ejector train operating.

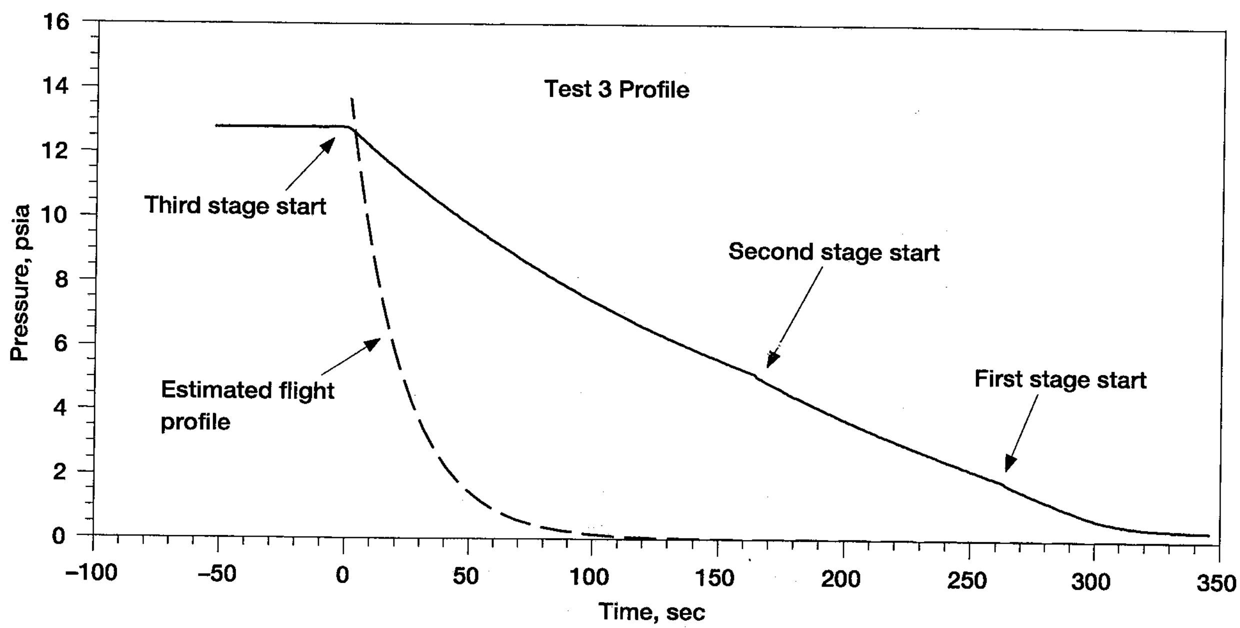

Figure 6.—Comparison of the ascent pressure profile simulation during Test 3 to the flight pressure profile.

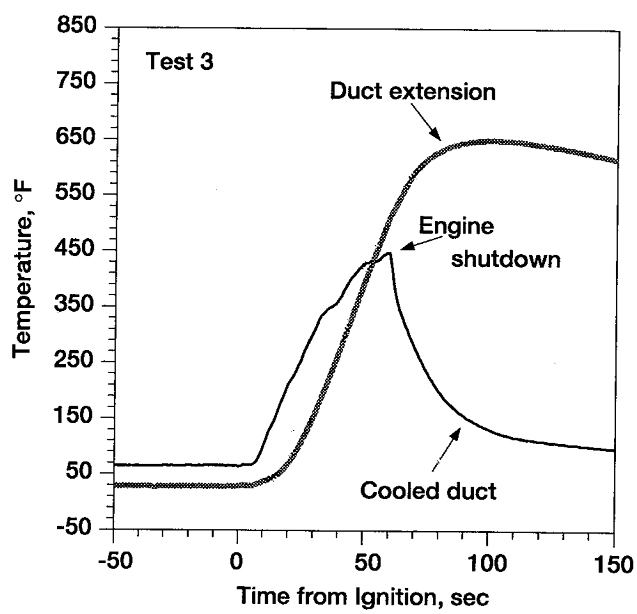

Figure 7.—Highest exhaust duct wall temperatures measured during the first X-Stage firing.

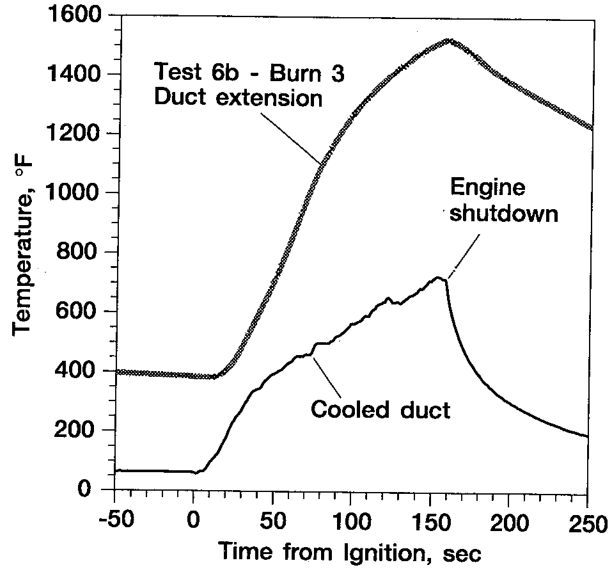

Figure 8.—Highest exhaust duct wall temperatures measured during the third firing of Test 6b.

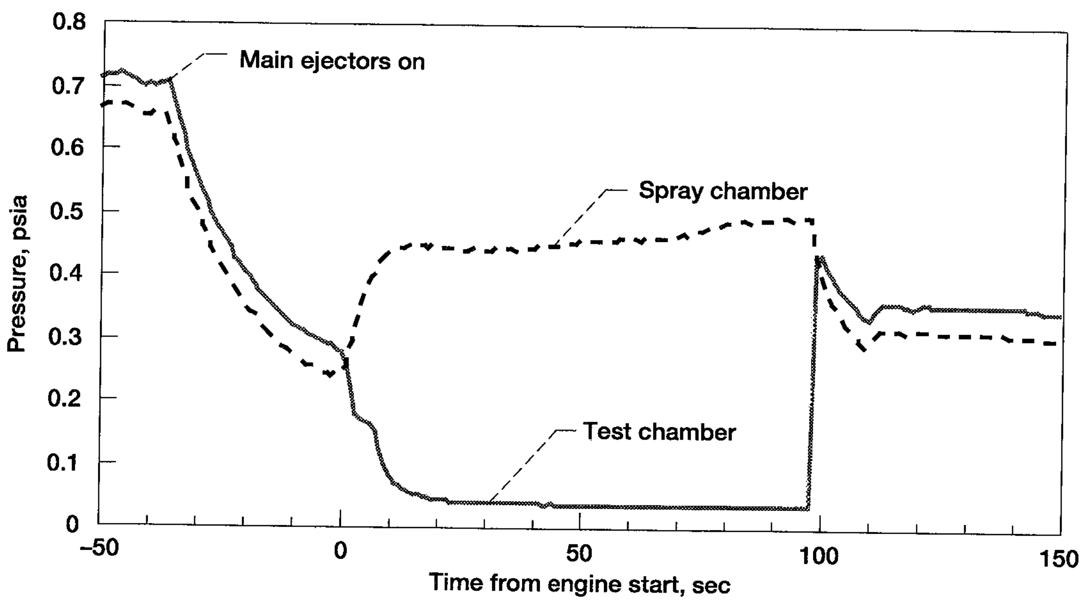

Figure 9.—Test and spray chamber pressures during an engine firing.

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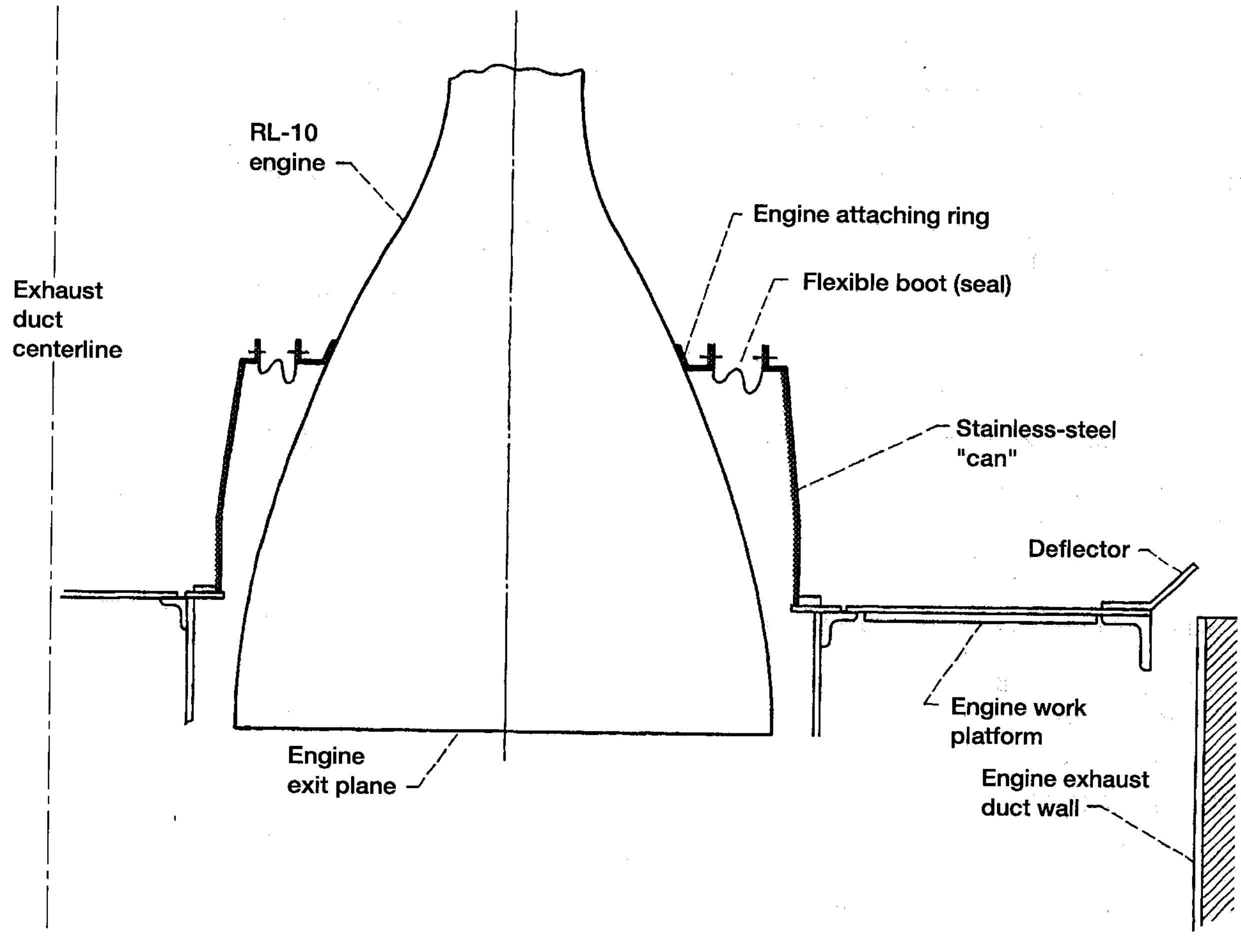

Figure 10.—Engine to work platform seal detail and deflector plate used during Centaur tests in B-2 facility.

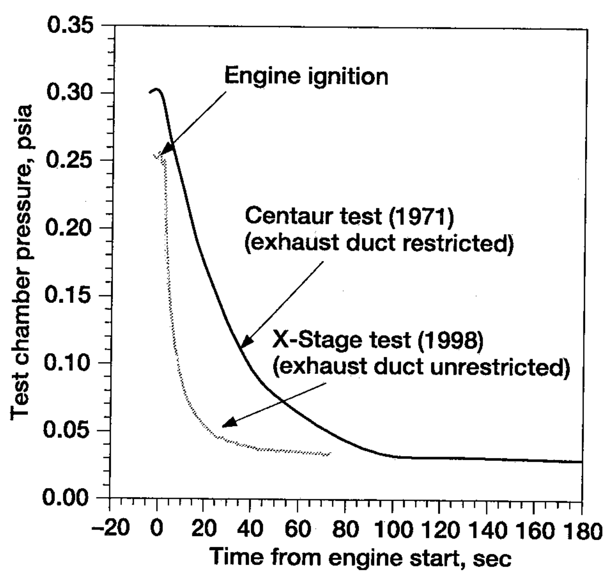

Figure 11.—Comparison of the test chamber pump down after engine ignition for the Centaur and X-Stage test configurations showing the effect of restricting the exhaust duct flow area.

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13. ABSTRACT *(Maximum 200 words)*

The Spacecraft Propulsion Research Facility at the NASA Lewis Research Center's Plum Brook Station was reactivated in order to conduct flight simulation ground tests of the Delta III cryogenic upper stage. The tests were a cooperative effort between The Boeing Company, Pratt and Whitney, and NASA. They included demonstration of tanking and detanking of liquid hydrogen, liquid oxygen and helium pressurant gas as well as 12 engine firings simulating first, second, and third burns at altitude conditions. A key to the success of these tests was the performance of the primary facility systems and their interfaces with the vehicle. These systems included the structural support of the vehicle, propellant supplies, data acquisition, facility control systems, and the altitude exhaust system. While the facility connections to the vehicle umbilical panel simulated the performance of the launch pad systems, additional purge and electrical connections were also required which were unique to ground testing of the vehicle. The altitude exhaust system permitted an approximate simulation of the boost-phase pressure profile by rapidly pumping the test chamber from 13 psia to 0.5 psia as well as maintaining altitude conditions during extended steady-state firings. The performance of the steam driven ejector exhaust system has been correlated with variations in cooling water temperature during these tests. This correlation and comparisons to limited data available from Centaur tests conducted in the facility from 1969 - 1971 provided insight into optimizing the operation of the exhaust system for future tests. Overall, the facility proved to be robust and flexible for vehicle space simulation engine firings and enabled all test objectives to be successfully completed within the planned schedule.

14. SUBJECT TERMS
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